LOW THRUST LIQUID ENGINES OF ISRO

N.SRIDHARAN DHAS, K.L.VALLIAPPAN
E.S.JOSEPH & C.G.BALAN

LIQUID PROPULSION SYSTEMS CENTRE
ISRO
INDIA

Abstract

The Liquid Propulsion Systems Centre (LPSC) of the Indian Space Research Organisation (ISRO) has developed and qualified low thrust bi-propellant and mono-propellant engines for use in Geostationary and Polar Orbiting Spacecrafts. The 440N thrust bi-propellant Liquid Apogee Motor (LAM) is used in INSAT-2 series of spacecrafts for orbit raising and the 22N thrust bi-propellant engines are used for attitude control and station keeping. For the Polar Orbiting Indian Remote Sensing Spacecrafts, attitude control is provided by the 1N thrust Hydrazine engines and orbit correction is provided by a 10N thrust Hydrazine engine. Currently 10N bi-propellant engines are being qualified for use in the GSAT and INSAT-3A class of spacecrafts slated for launch within the next two years. This paper briefly presents the configuration, design, development and qualification programme and performance parameters of these low thrust engines.

Introduction

The Liquid Propulsion Systems Centre (LPSC) initiated development activities related to bi-propellant (MON-3, MMH) and mono-propellant (Hydrazine) thrusters in the early eighties to meet the propulsion needs of the indigenously built Remote Sensing and Multipurpose Communication Spacecrafts. Developmental efforts and the solutions found for the problems faced in the initial phase have yielded expertise in the design, fabrication and testing of small engines for spacecraft needs. Inhouse facilities for precision fabrication, clean assembly and environmental testing (including simulated high altitude performance evaluation) have been established. The 440N & 22N bi-propellant engines for Communication Spacecrafts and the 10N & 1N mono-propellant engines for Remote Sensing Spacecrafts are now under regular production. These engines have gone through extensive development/ qualification testing and have attained a high level of design maturity. This paper summarises in brief the configuration and design aspects of these thrusters. The extent of qualification testing and the performance achieved are outlined in this paper.

Propulsion System Description

A unified bi-propellant liquid propulsion system is used in the INSAT-2 series of spacecrafts for orbit raising and attitude control. Fig. 1 gives the propulsion system layout for the INSAT-2E. It has one 440N Liquid Apogee Motor (LAM) and 16 nos of 22N control thrusters. The oxidiser (MON3) and fuel (MMH) are stored in Titanium tanks provided with surface tension-type propellant management devices. The volume of each tank is 770 litres and the total propellant loaded is 1400 kgs. The pressurant gas (Helium) is stored in two gas bottles of 35.5 litres volume each at a pressure of 23.5 MPa. The high pressure Helium is regulated down to 1.6 MPa to pressurise the propellant tanks. The LAM engine operates in a regulated mode whereas the 22N control thrusters function in a blowdown mode. The control thrusters are connected in two groups of eight engines each for redundancy. Bi-propellant latch valves ensure isolation of any one group of control thrusters.

Proceedings of the Euro-Asia Space Week on Cooperation in Space – 'Where East & West Finally Meet', 23-27 November 1998, Singapore (ESA SP-430, February1999).

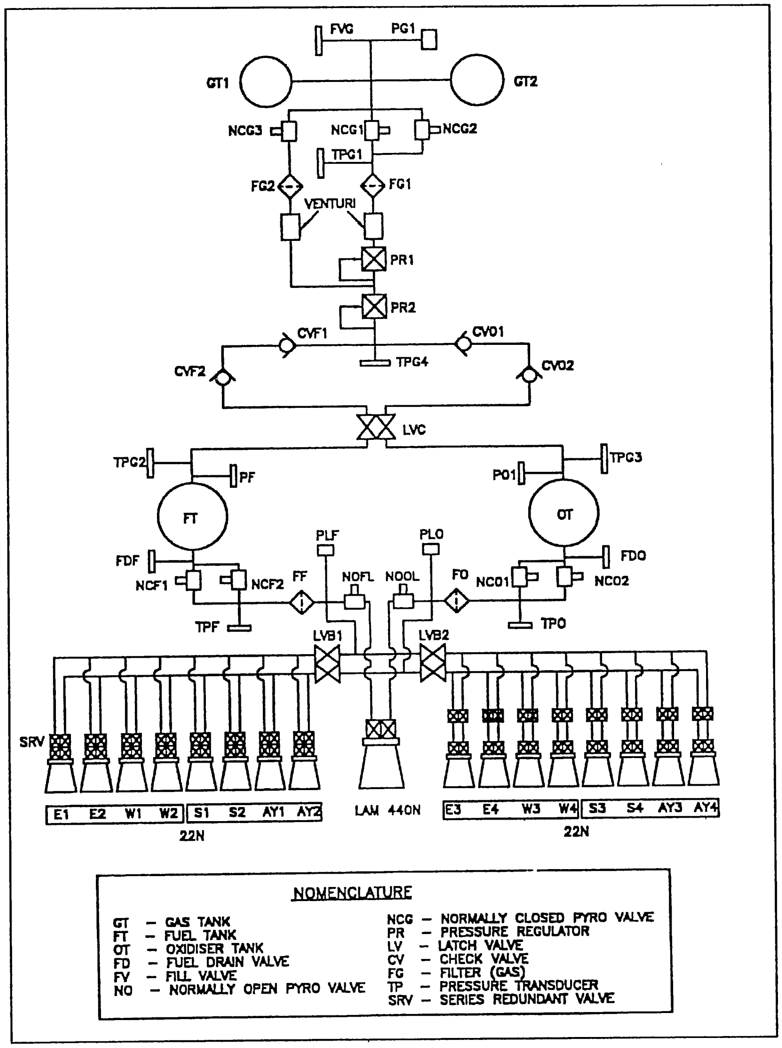


Fig. 1- INSAT 2E PROPULSION SYSTEM SCHEMATIC

Fig. 2 gives the layout of the Hydrazine system used in the Indian Remote Sensing Spacecraft IRS-P4. The propulsion system meets the functional requirement of achieving the required velocity to reach the final orbit taking into account the spacecraft injection errors and maintain the spacecraft in orbit for a nominal mission life of three years. Hydrazine is stored in four 30 litres volume spherical tanks having surface tension-type propellant management devices. The total propellant loading of 84 kgs. meets the required mission life. The system is pressurised with Helium initially to 2.4 MPa and the final blowdown pressure is 0.8 MPa. 16 nos. of 1N thrusters (grouped into two blocks of 8 each) provide attitude control and a single 10N engine is used for orbit raising manoevres.

Liquid Apogee Motor (LAM)

The LAM develops a thrust of 440N with a minimum deliverable specific impulse of 3041 N sec./kg mass. The mixture ratio of propellants (O/F by weight) is 1.65 and the nozzle area ratio is 160. The injector is a co-axial swirl single element type and is made of Titanium (Ti6Al4V). The thrust chamber is made of Columbium alloy, silicide coated and radiation cooled.

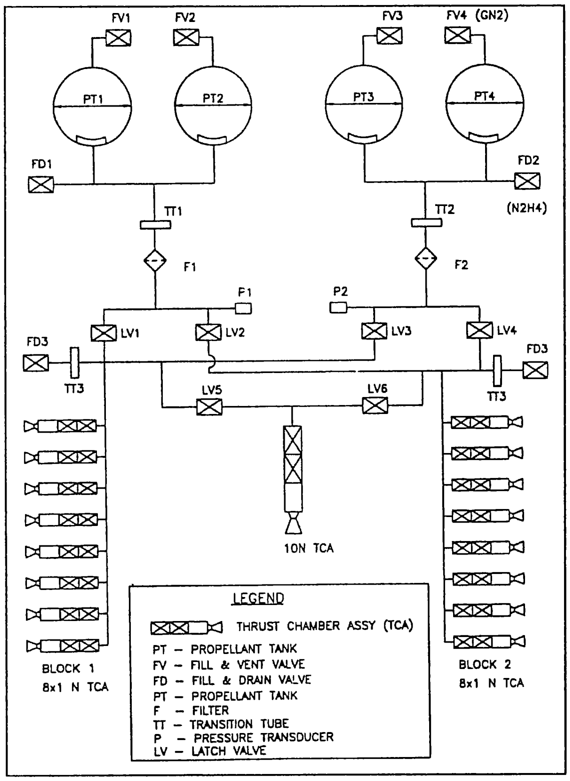


Fig. 2 - IRS-P4 PROPULSION SYSTEM SCHEMATIC

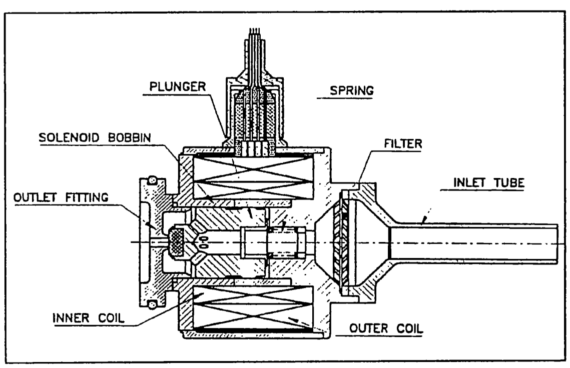


Fig. 3 - LAM SOLENOID VALVE

The injector assembly is electron beam welded to the thrust chamber. The nozzle divergent is also made of Columbium alloy, radiation cooled and silicide coated. The valves used are of the solenoid type with sliding plunger. The valve sealing is achieved by a flat soft seal crimped in the armature loaded against the flat stainless steel seat by a helical spring. The solenoid features dual redundant coils potted and hermetically sealed. A pleated disc filter of 40 microns (abs.) is installed at the valve inlet. The valve has an all welded construction. Fig. 3 gives the cross section of the valve used in the LAM. Fig. 4 shows the LAM engine assembly. The weight of the LAM assembly (including the valves) is 4 kgs.

The LAM development program was carried out in two phases – sea level version development and flight version development. In the preliminary design validation tests (sea level), seven engines were tested where the longest steady state firing was for 6000 sec. and the maximum cumulative firing in a single engine was for 11700 sec. The flight version development and qualification tests were carried out in a High Altitude Test Facility. Three engines were tested as part of the development tests and five engines as part of the qualification tests. All the hardwares were taken through a steady state firing of 3000 sec. and one of the engines was tested for a cumulative firing of 23542 sec. under the qualification program. Single steady state firing in the High Altitude conditions is limited to 3000 sec. due to facility constraints. The qualification test summary is given in Table 1.

Hard ware	Cumulative burn time in sec.	Thermal cycles/ restarts	Propellant Temp. range (Deg. C)	OFF-NOMINAL CONDITIONS	
				Injection pressure variation (bar)	MR Variation
1	9,500	24	-1 to 35	$\pm$2 bar	1.42 to 1.85
2	9,544	25	-1 to 62	$\pm$2 bar	1.45 to 1.81
3	9,588	23	4 to 67	$\pm$2 bar	1.41 to 1.90
4	9,565	16	ambient	2 bar blow down	1.6 to 2 blow down
5	23,542	24	ambient	$\pm$2 bar	1.43 to 1.97

Table 1 - LAM QUALIFICATION TEST SUMMARY

The LAM has demonstrated very good performance for various injection pressures in the range of 0.9 MPa to 2 MPa, mixture ratios in the range of 1.2 to 2, propellant temperatures in the range of 0°C to 65°C and supply voltages in the range of 28VDC to 42VDC. Plots of specific impulse as a function of injection pressure and mixture ratio are included in Figs. 5 and 6.

Table 2 presents the on-orbit performance of INSAT-2 LAM.

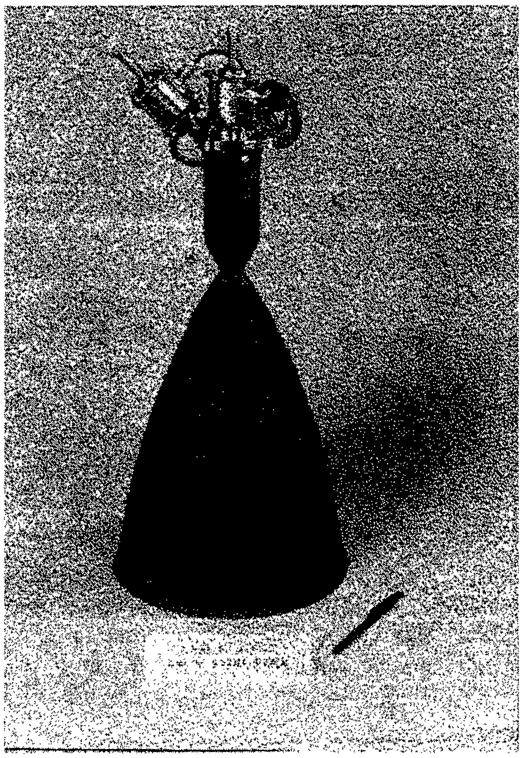


Fig. 4 - LAM ENGINE ASSEMBLY

Mission	Largest steady state firing duration (sec.)	Total burn time (sec.)	Avg. Isp (N sec./kg mass)
INSAT-2A	3,529	4,740	3061
INSAT-2B	3,463	5,130	3051
INSAT-2C	3,842	5,796	3021
INSAT-2D	4,033	5,783	3080

Table 2 - 440 N LAM ON-ORBIT PERFORMANCE

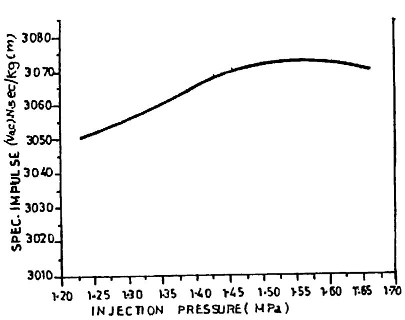


Fig. 5 - LAM PERFORMANCE
SPECIFIC IMPULSE Vs INJECTION

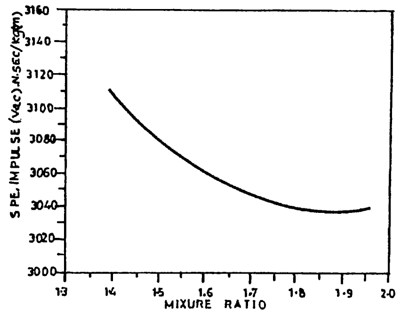


Fig. 6 - LAM PERFORMANCE
SPECIFIC IMPULSE Vs MIXTURE RATIO

22 N Engine

The 22N engine also uses co-axial swirl type Titanium alloy injector welded to a silicide coated Columbium thrust chamber. The thrust chamber is both film and radiation cooled. The nozzle area ratio is 100. This engine operates at 0.68 MPa nominal chamber pressure and the specific impulse in continuous operation is 2780 N.sec./kg.mass. In pulse mode operation the minimum impulse bit obtained is 65 mN sec. for 8 ms operation. The weight of the 22 N engine (including valves) is 0.8 kg.

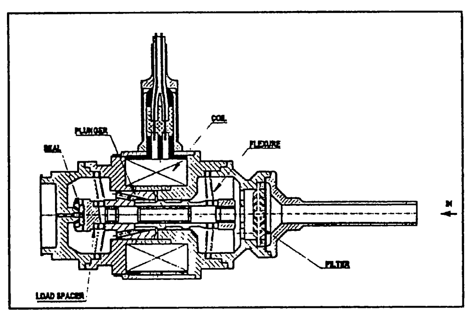


Fig. 7 - 22 N ENGINE VALVE

Fig. 7 gives the 22N engine valve configuration. The thruster valves are solenoid types with plunger type armature and the valve sealing is effected by a teflon disc in the plunger pressing against a stainless steel seat. All moving elements - armature and the poppet seal assembly- are mounted on a set of four flexures which provide friction-free guidance to the armature, in addition to the seat load necessary for meeting the leak rate specification. The solenoid coil is potted and hermetically sealed. A pleated disc filter of 40 micron (abs.) rating is provided at valve inlet. The valve features an all welded construction. This

valve has undergone extensive development and qualification programme in both the component and thruster levels involving over 30 hardwares. These tests include functional, environmental and also wet cycling test for one million operations.

After the development programme was completed, 10 numbers of 22N engines have undergone the qualification programme. As the engines are meant for pulse mode operation, the performance of the engine is evaluated for various pulse widths and duty cycles for a wide range of injection pressures (0.9 MPa to 1.9 MPa) mixture ratios (0.2 to 2), propellant temperatures (-5°C to 65°C) and valve supply voltages (28 VDC to 42 VDC). Thrusters are qualified for 0.3 million pulse operation in various pulse widths and duty cycles. The 22N engine has demonstrated the capability to withstand a cumulative firing duration of 70,000 sec. and maximum continuous steady state firing of 10,000 sec. Fig. 8 shows the 22N engine.

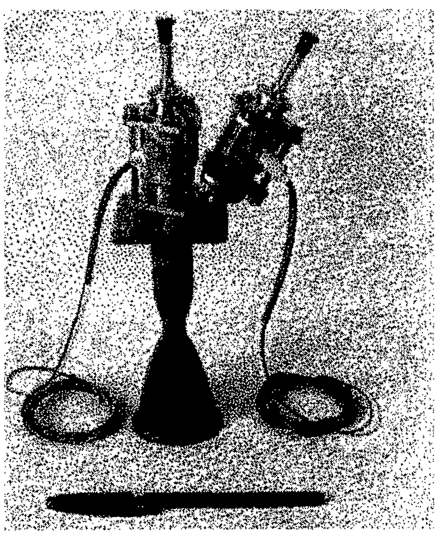


Fig. 8 - 22 N ENGINE

For the GSAT satellite attitude control and station keeping, 10N bi-propellant engines will be used. These engines are under development and have a configuration similar to that of the 22N engines. In these engines, solenoid valves with clapper-type armature are used for flow control. These valves can meet the flow requirements of 22N engines also. Fig. 9 presents the 10 N engine valve configuration.

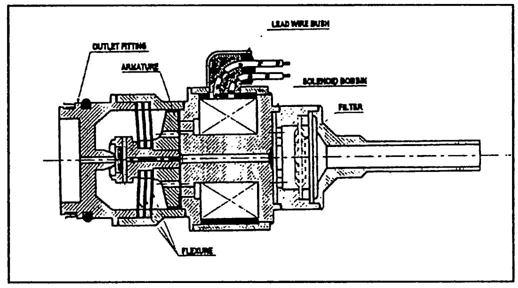


Fig. 9 - 10 N ENGINE VALVE

For the 22N/10N engines in future satellites, control valve assembly with the latch valve located upstream of the solenoid flow control valve (Fig.10) is under development and qualification. With these valves any faulty thruster in the system when detected, can be isolated.

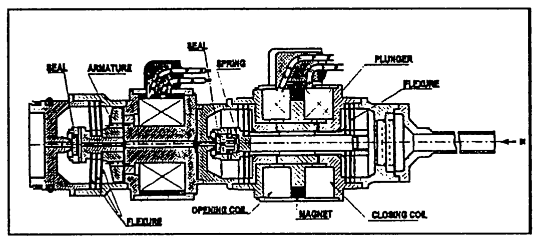
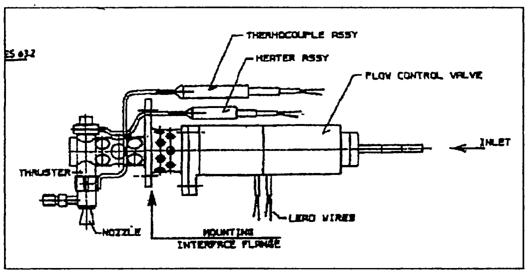


Fig. 10 - CONTROL VALVE ASSEMBLY WITH UPSTREAM LATCH VALVE

Mono-Propellant Hydrazine Thrusters

LPSC has developed and qualified 1N &10N Hydrazine catalytic thrusters for the propulsion system of Indian Remote Sensing Spacecrafts. Fig. 11 shows the configuration of the 1N engine. The weight of the 1N engine (including valves) is 0.35 kg. Thruster assembly consists of Hydrazine catalytic thruster, series redundant on/off valve and the catalyst bed heater. The thermal stand off, injector, reaction chamber and nozzle are the main elements in the Hydrazine catalytic thruster. The reaction chamber contains an Iridium based catalyst packed in a high temperature alloy chamber and retained within by a retainer mesh. Hydrazine from the on/off valve is injected into the reaction chamber by an injector capillary tube running from the valve – thruster interface flange into the reaction chamber. Hydrazine from the injector capilliary is distributed equally over the catalyst bed.



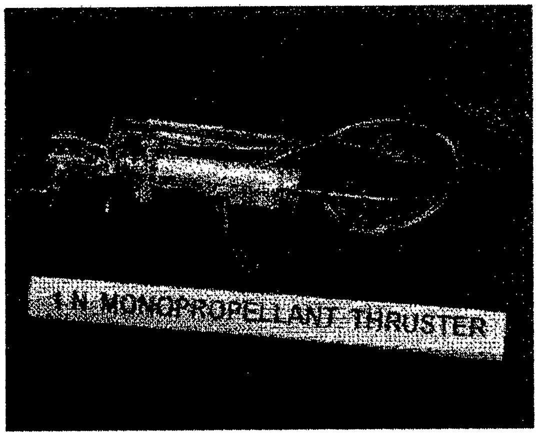


Fig. 11

A set of two normally closed, solenoid valves connected in series, control the Hydrazine flow into the engine either in pulse mode or continuous mode. The seals are made of Ethylene Rubber which has demonstrated long term compatibility with Hydrazine. Catalyst bed heating (min.temp required : 393 K) for long life and repeatable performance is maintained by a catalyst bed heater which has a Nichrome heater element in a hermetically sealed cover. The catalyst bed temperature while the heater is on and during the thruster firing operation is monitored by clamping a Chromel Alumel type thermocouple on the reaction chamber.

The reaction chamber with an area ratio of 50 is brazed to the thermal standoff which provides structural support and thermal isolation to the on-off valve. In order to minimise the heat transfer from the reaction chamber to the environment, the thruster is gold plated and enclosed in a gold plated shield.

The thruster produces 1 N nominal thrust at 1.8 MPa, 1.35 N at 2.4 MPa and 0.35 N at 0.8 MPa injection pressure. The minimum impulse bit produced is 30 mN sec. with repeatability within 10%. The steady state operation (single burn) specification is 10,000 sec. (max). This thruster has demonstrated a cycle life of 26,00,000 pulses, a cumulative steady state firing of 24000 sec. and 40 starts at +5°C.

1	a) 1,60,000 hot pulses, 5 to 95% duty ĉycle b) 24,000 sec. cumulative steady state firing (include 1x5,000 sec. continuous steady state firing) c) 10,000 sec. continuous steady state firing in one thruster. d) 40 cold starts	24, 18 & 11 bar	6 Nos of thrusters
2	a) 1,60,000 hot pulses, 5 to 95% duty cycle b) 24,000 sec. cumulative steady state firing (include 1x5,000 sec. continuous steady state firing). c) 40 cold starts. d) Extended pulse mode firing of 5 lakh pulses	24, 18 & 11 bar	1 thruster
3	a) 1,60,000 hot pulses b) 24,000 sec. cumulative steady state firing c) 40 cold starts d) Extended pulse mode firing of 26 lakh pulses	24, 18 & 11 bar	1 thruster

Table 3 - QUALIFICATION TEST FIRING SUMMARY 1N HYDRAZINE ENGINE

After the development program, 8 nos of 1N thrusters were subjected to elaborate qualification test firing as in Table 3. The series redundant valves were separately qualified for functional and environmental specifications

including thermo vacuum test, life cycling test and compatibility test (more than 3 years with Hydrazine in one valve).

The 10 N Hydrazine thrusters are developed and qualified for orbit raising operations in Remote Sensing Spacecrafts. The general configuration of 10 N engine is similar to that of the 1 N engine except for the injector. In the 10 N engine, the injector capillary is distributed equally over the catalyst bed by a diffuser with three point injectors fixed at the end of the injector tube. The 10 N thruster is qualified for a cumulative duration of 45000 sec. in steady state operation with injection pressures varying from 2.4 MPa to 0.8 MPa.

Development tests carried out on five thrusters have been followed by successful qualification tests on three thrusters. A Hydrazine throughput of 110 kgs. has been established in steady state burn with repeated single burns of 1000 sec. duration. Specific impulse better than 2158 N sec./kg mass has been realised.

Conclusion

The efforts taken by ISRO from the early eighties towards the development of low thrust, high performance engines for spacecraft propulsion have resulted in achieving a high level of design maturity. The utilisation of more than 150 such engines in the INSAT satellites (Geostationary orbit) and IRS satellites (Polar Orbit) reflects the significant in-house capability to realise both mono- propellant and bi-propellant engines in the 1 N to 440 N thrust range. The production costs of these engines turnout to be very attractive in comparison to other engines available elsewhere.

Some of the on-going activities in LPSC include the improvement in the specific impulse of the LAM engine, qualification tests of the 10N bi-propellant engine and the enhancement of propellant throughput performance of the 10 N Hydrazine engine.

Acknowledgement

The authors wish to thank Mr.G.Madhavan Nair, Director, LPSC for the encouragement he has given in bringing out this paper. The authors gratefully acknowledge the unceasing support being given by Dr.K.Kasturirangan, Chairman, ISRO to the spacecraft propulsion area. The authors also wish to acknowledge the contributions of the design and development engineers M/s. M. Neelakanta Reddy, C. Amarasekharan, B. K. Venkataramu, Raju.P.Thomas and S.Venkateswaran of LPSC, ISRO. Without their efforts, this programme would not have been the success which they helped make possible.